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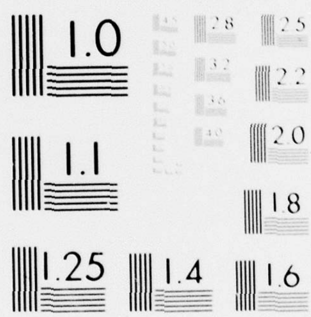

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ARMY HELICOPTER FLIGHT SIMULATION, A RESEARCH AND DEVELOPMENT TOOL FOR THE FUTURE

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INTRODUCTION

In the last decade, the United States Army has become increasingly interested in flight simulation of its helicopters. In the early 1960's, the Army purchased the 2B24 flight simulator trainer to replace the antiquated 1CA1 "Link Trainer". Subsequently, the Army has purchased flight trainers to simulate the CH-47 and AH-1Q helicopters. However, all these simulators were used primarily for pilot training. It was not until recently that the Army has become interested in piloted simulation as a development tool. Piloted simulations are part of the Army's XV-15 Tilt Rotor Aircraft program and the Rotor System Research Aircraft (RSRA) program. In both of these programs, the simulation was used as a verification tool, i.e., verify handling qualities of the helicopter after design.

This paper presents the results of a piloted simulation that was used as a means of determining the cause of an Army 214 Helicopter accident. The simulation program was also used to augment the design of a flight control monitor system and to investigate the effects of possible control system changes to the 214 helicopter.

ACCIDENT BACKGROUND

In March 1976, the United States Army Aviation Engineering Flight Activity (USAAEFA) was directed by the United States Army Aviation Command (USAVSCOM) to test the Airworthiness and Flight Characteristics of the Iranian Model 214C Helicopter. It was during this testing that the helicopter experienced excessive main rotor flapping, and subsequently, the helicopter disintegrated in flight.

A post crash fire destroyed all in flight data and recordings, thus making it impossible to determine the cause of the accident. However, an observer in the chase helicopter stated that the helicopter was in a nose down and left roll attitude prior to its disintegration. An inspection of the wreckage disclosed that the main rotor shaft, hub, and blades had experienced excessively high main rotor flapping prior to their separation from the helicopter.

During this particular flight, the crew was testing the failure modes of the helicopter's Automatic Flight Control System (AFCS) for compliance with paragraph 3.5.9 of reference 1. The test helicopter was configured so that the flight engineer could induce a hardover failure in a given AFCS channel by depressing the respective button on a special "Hardover Test Box". Upon releasing the button, the hardover was removed and the AFCS resumed its normal function. It was hypothesized that the flight engineer inadvertently released the hardover button coincident with the pilot's corrective input, consequently resulting in a hardover in the other direction.

The primary objectives of the simulation were to duplicate all possible AFCS failures, both inadvertent and advertent, inconjunction with subsequent pilot actions that could have caused the accident. Also, we wanted to try and identify the best pilot technique to use to recover the helicopter from an AFCS hardover.

FACILITY DESCRIPTION

The Flight Simulator for Advanced Aircraft (FSAA) is a piloted simulator at NASA Ames Research Center that has six degrees of freedom. The motion system is capable of $\pm$ 5 feet displacement in the vertical direction, $\pm$ 4 feet in the longitudinal direction, and the unique capability of $\pm$ 50 feet of lateral displacement. The FSAA can achieve velocities of 8.65 ft/sec, 6.32 ft/sec and 17.00 ft/sec in the vertical, longitudinal and lateral directions respectively. Figure one is a comparison of maximum angular rates of the FSAA and maximum rates of the 214C helicopter measured during flight testing. It should be noted that none of the 214C helicopters' rates exceed the capability of the simulator. If the actual helicopter rates exceeds the simulator's capability, false pilot cues would result which would limit/degrade the overall simulation.

The total Flight Simulator for Advanced Aircraft (FSAA) system is depicted in figure two. A Xerox Data System (XDS) Sigma 7 and an XDS Sigma 8 are used to provide driving values for the simulator hardware inconjunction with processing feedback signals from these units. The digital computers communicate with each other via

the XDS 7908 interface hardware. Analog to digital converters, digital to analog converters, and discrete input/output channels are used to process computer generated signals to simulation hardware through the instrument rack. The simulation is conducted from the simulation control room which is the basis for operating/debugging the computer system. Signals are sent to the Visual Flight Attachment, which generates the pilot's visual scene, from the instrument rack. Signals are sent through the Drive Rack to the Motion System Drive which drives the Cockpit to the desired position and attitude.

MAXIMUM RATES (DEG/SEC)		
AXIS	214C	FSAA
DIRECTIONAL	20.4	40.1
LATERAL	28.6	40.1
LONGITUDINAL	16.2	101.4

FIGURE 1. THE FSAA MAXIMUM ALLOWABLE RATES COMPARED TO THE MAXIMUM MEASURED RATES OF THE 214C HELICOPTER

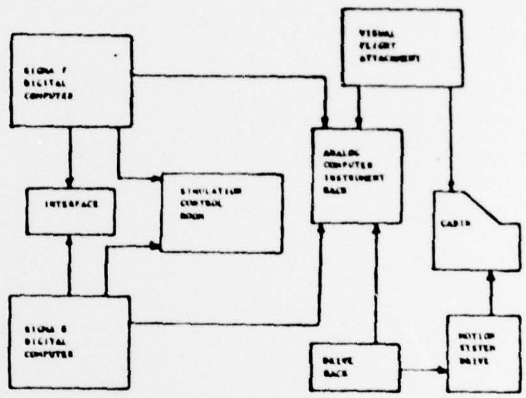

FIGURE 2 THE FSAA DUAL PROCESSOR SYSTEM SCHEMATIC

HELICOPTER MATHEMATICAL MODEL

The requirement to simulate AFCS hardover failures and subsequent recovery techniques demanded that the mathematical model be capable of accurately describing large pertubations from trimmed flight conditions. The mathematical description of the 214 helicopter had to have a high degree of exactness in the calculation of the rotor blade flapping because of the suspected excessively large flapping during the accident. The mathematical model is a non-linear, highly coupled formulation in six degrees of freedom. The math model was a variation of the math model used in the Bell Helicopter Textron/Army/NASA XV-15 Tilt Rotor Program Simulation. The main rotor and tail rotor performance and dynamics were calculated utilizing a lifting disc model. The lift and drag forces for the helicopter fuselage and tail surfaces were calculated from aerodynamic functions stored in the computer program. These aerodyamic functions were derived from wind tunnel test. The math model also represented the engine power and engine governing characteristics. Further description and explanation of the math model can be found in reference 2 and 3.

The Automatic Flight Control System (AFCS) and its associated logic and control authority was also included in the Math Model. The SCAS logic system also included provisions for system failures, i.e., hardovers. In order to perceive the true impact of the simulation program, a brief description of the 214 helicopter's flight control system is in order. The AFCS system of the 214 helicopter consists of a Stability and Control Augmentation System (SCAS) and the Attitude Retention System (ATTD). The SCAS is integrated into the cyclic and anti-torque control systems to automatically provide improved stability and handling qualities of the basic helicopter. SCAS input to the control system is accomplished by an electro-hydraulic servo actuator installed series with the control system. The ATTD mode automatically maintains the desired pitch and roll attitude and heading of the helicopter. The attitude inputs to the control system are accomplished through combined inputs from the SCAS series electro-hydraulic actuators and an electro-mechanical trim actuator connected in parallel.

The complex nature of the nonlinear coupled mathematical equations makes their computation quite lengthy, and could compromise a real-time simulation. The real-time cycle rate has been a considerable problem when doing pilot in the loop simulation in the past. The integrity of the simulation is heavily dependent on the computer program cycle time. Cycle time is the length of time for the computer to integrate all solutions of the differential equations and provide the respective inputs to the simulation hardware, i.e. motion drive, visual system, etc. Generally the solution of the nonlinear equations results in the most significant increase in cycle time.

The unique aspect of this simulation, as opposed to other flight simulations, was that two digital Sigma 7 and Sigma 8 computers functioned in parallel to solve the math model equations. This dual-processor was initiated in the simulation of the RSRA helicopter. (4) The dual-processor was necessary in order to obtain an acceptable cycle time of approximately 40-50 milli seconds. The cycle time would increase by 40 percent if only one Sigma 8 computer was used in lieu of the dual-processor system. This would result in a marginal cycle time.

SIMULATOR CORRELATION AND VERIFICATION

Correlation of the various elements of the helicopter mathematical model were carried out after the completion of the programming of the computers. The primary purpose of the correlation was to determine the degree of fidelity the simulation had with the actual

214 helicopter. The simulator correlation and verification, as required in reference 5 test plan, was done at the airspeed, center of the flight conditions prior to the accident. The evaluation and verification was predicated on comparison of flight test data of references 6, 7, and 8 with the equivalent simulator results. The simulator evaluation consisted of the following test a) Rotor Performance, b) Control system characteristics, c) Control positions in trimmed forward flight, d) Dynamic Stability, e) Controllability, and f) Automatic Flight Control System (AFCS) failures.

Rotor thrust and power calculations were obtained by effectively isolating the rotors from the fuselage aerodynamic effects and using these results to compare with the results from BHT AGAJ-7407 (C81) computer program. Any variation in performance between the simulation and C81 was rectified by making slight alternations to the basic aerodynamic functions in the simulator program. This was a performance match of the simulator lifting disc math equations, with the C81 rotorcraft analysis.

The control system characteristics were checked to insure that the breakout forces, trim rates, force gradients, control travel and trim authority matched those of the 214 helicopter. The control positions in trimmed forward flight of the FSAA were checked against flight test results. Figure 3 shows the close correlation of control position locations in trimmed forward flight. The static lateral-directional stability and the dynamic stability of the FSAA were checked against flight test data. The results showed good correlation.

Controllability tests were conducted to evaluate the control power, control response, and control sensitivity of the simulator. Testing was accomplished by applying step control inputs in increasingly equal increments in each direction about the pitch, roll, and yaw axis. Figures 4 and 5 depict the results of the controllability test of the simulator compared to flight test data. The fidelity of the simulator is verified by the close correlation of the controllability plots. This infers that for a given control input, the rotor/airframe coupling is equivalent to that of the actual aircraft.

Automatic Flight Control System (AFCS) failures were conducted to verify that the simulated AFCS logic operated properly and that the simulated hardover failures were representative of the actual helicopter. Figure 6 is a time history plot of comparative hardover failures. Observing the time histories, it should be noted that the cyclic stick's movement is in the right direction. The

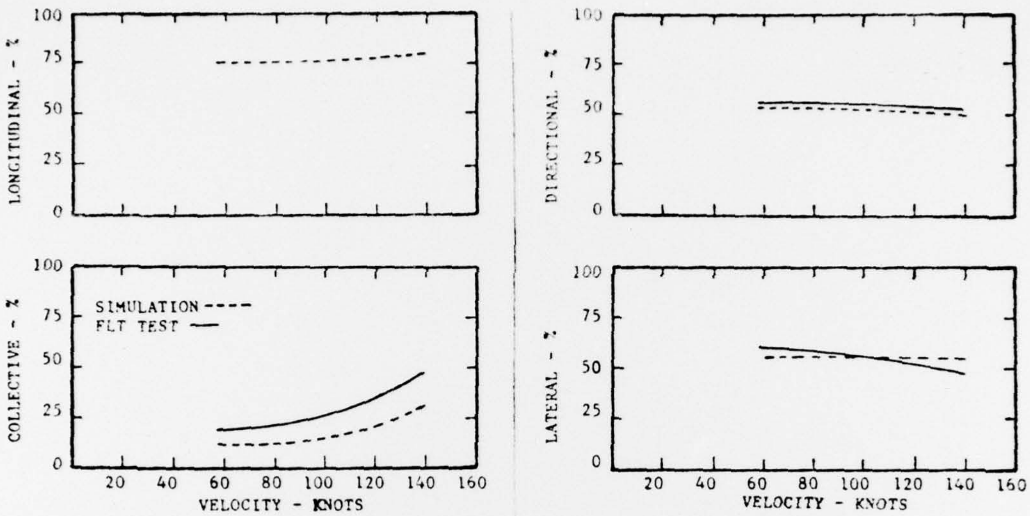

FIGURE 3. CORRELATION OF STATIC TRIM PARAMETERS

pitch rate and attitude show simular trends between flight test and simulator data. It is difficult to match the parameters in a hard-over test because they are all dependent on the pilot delay time. It was almost impossible to have the simulator pilot delay the exact time as on the flight test results. Not shown are other pilot inputs such as lateral cyclic stick, directional and collective stick position which have a slight effect on the parameters displayed. The figure shows minimal discrepancy between actual main rotor flapping and the calculated main rotor flapping of the simulator. With the major discrepancy being credited to diffeence in pilot fore and aft cyclic stick movement as opposed to a shortcoming attributable to the math model. In general, data is strongly influenced by pilot inputs, consequently making simulated hardover verification extremely difficult.

TEST METHODOLOGY

The test consisted of a pilot flying the simulator to accom-the Iranian Helicopter were twofold. The first was to investigate and determine the possible factors and sequence of events leading to high main rotor flapping during maneuvers. The second was to investigate various pilot techniques of preventing flapping and consequently mast bumping during recovery from unusual attitudes.

The test consisted of a pilot flying the simulator to accomplish the above objectives in accordance with the established test

plan. (5) During the simulated flight, hardover failures, both single and dual axis, were induced about each aircraft axis, i.e., pitch, yaw and roll. The majority of the simulator flights were conducted at the flight conditions that were believed to be those of the helicopter at the time of the accident. These flights are referred to as baseline flights. The baseline configuration was 13000 pounds gross weight, with center of gravity at the aft limit of 144 inches, an altitude of 5000 foot and an airspeed of 120 knots.

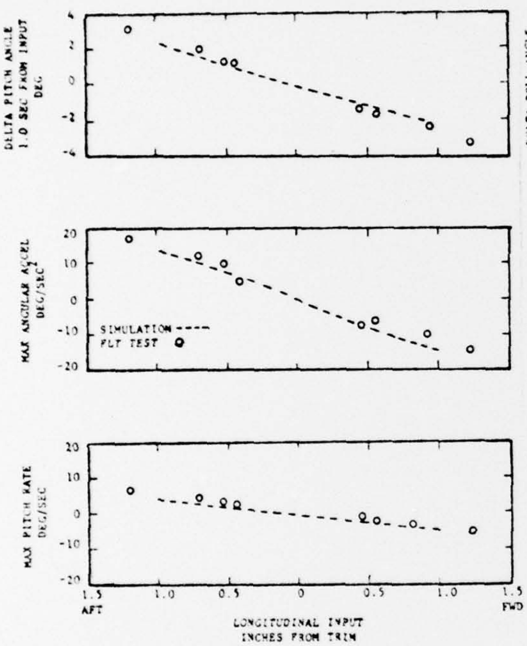

FIGURE 4. LONGITUDINAL CONTROLLABILITY COMPARISON

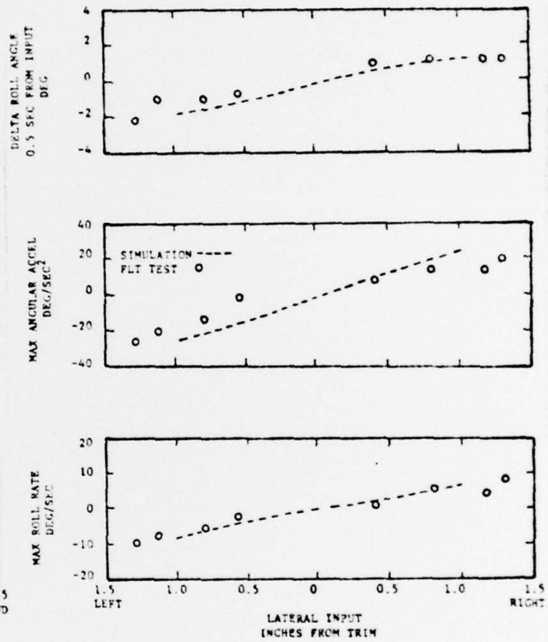

FIGURE 5. LATERAL CONTROLLABILITY COMPARISON

Data was obtained during hardover failures with the pilot delaying his corrective inputs in one second inputs until the maximum delay time was achieved. During flight test programs, hardover delay times are generally built up from a minimum time to a maximum pilot delay time was achieved. During flight test programs, hardover delay times are generally built up to from a minimum time to a maximum delay time. Previous data shows that the maximum pilot delay times in the simulator are in close agreement with results from flight test programs contrary to the thought that the lack of pilot anxiety in a simulator would increase the delay times. (9) However, it is thought that pilot training from flight test experience carried over to the simulator contributing an inhibiting effect. In addition, the pilot has a very low motivation to achieve long delays in an aircraft or simulator. The maximum delay period was determined as being the delay time when

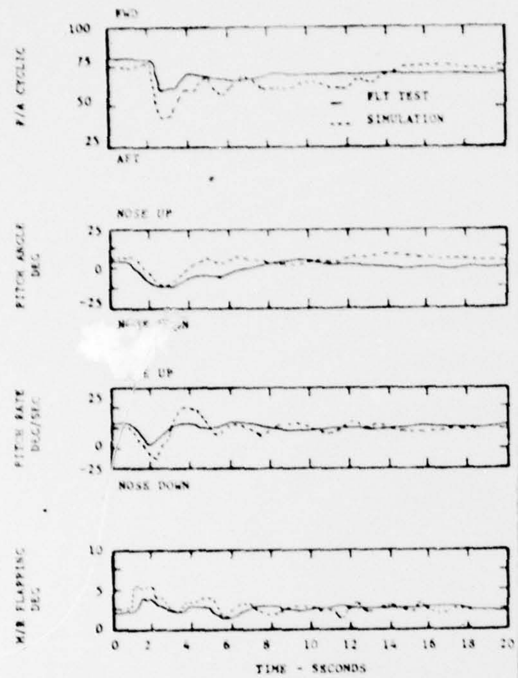

FIGURE 6. PITCH DOWN HARDOVER TIME HISTORY

TABLE I

PILOT RECOVERY TECHNIQUES	
#1	After the initiation of the hardover, the pilot delays the appropriate time and returns helicopter to a level attitude.
#2	Same as technique #1 except that the hardover is removed at the start of recovery.
#3	Same as technique #1 except that the AFCS was disengaged at the beginning of the pilot recovery.
#4	Same as technique #1 except that the collective was dumped during recovery to simulate inadvertent power drop.
#5	After the initiation of the hardover, the pilot delays the appropriate time and arrests the rate/attitude change and then flies out of the attitude in a gentle maneuver.

100 percent main rotor flapping was reached and/or the pilot is still able to regain control of the aircraft.

After the delay, the pilot would use one of five pilot technique (See Table I) to correct for the hardover condition and regain control of the aircraft. One criteria for selecting the pilot techniques investigated were to select those on the basis of being a possible series of events leading to the 214 accident. It was also imperative not only to specify the technique that directly lead to the accident, but also the technique that would result in the minimum main rotor blade flapping during recovery. The latter technique would become the established emergency procedure. Pilot technique number 2 was thought to be the most severe from the aircraft attitude and main rotor flapping aspect. (In technique #2 by removing the simulated hardover at the time of pilot recovery, the AFCS comes back on line and senses the large attitude rate, immediately compensates by moving the actuator just as the pilot is making his inputs. This is in all actuality a hardover in the reversed direction from the original hardover). For each individual pilot technique and time delays, various parameters, e.g., helicopter attitude, control

positions airspeed, etc., were recorded with particular monitoring interest on main rotor flapping.

A total of four test pilots participated in the program. Of the four pilots, three were Army test pilots with a wide background in helicopter testing and one pilot was a Bell Helicopter test pilot with previous simulator experience and 214 helicopter test time. Of the three Army test pilots, none had previous simulator experience other than training flights in the UH-1H Army trainer 2B24. One Army pilot had recently completed a flight test program on te 214C helicopter, and consequently, was quite familar with its performance and handling qualities.

Certain baseline flights were repeated with all pilots utilizing all pilot recovery techniques to insure adequate data overlap. With the comparison of the simulation parameters, one was able to ascertain that the effects of different pilots on the test data was minimal. After completion of the baseline testing, hardover testing was done in a combination of axis concurrently. This required the simulation engineer, who was stationed in the simulation control room, to induce simultaneous failures in two different axis, example: pitch down and left roll. This was performed to determine the pilot effort required to control the dual hardover failures versus single failures.

Upon completion of the above hardover testing, the basic test configuration was varied for the remainder of the test. The airspeed was varied from 100 to 160 knots along with varying the c.g. from aft to forward and varying altitude.

The prepare the simulator to run the different test cases, all that was necessary was to make an input change to the computer program which did not take much time. This is a demonstration of the flexibility that a simulator program has over a flight test program, which would have required extensive work and effort to change the flight conditions.

Testing was also done under simulated IFR conditions. This was accomplished by having the computer generate and project a complete cloud cover over the pilot's visual screen. This required the pilot to completely regain control of his aircraft after a hardover failure by use of the flight instruments only. This test was not performed as part of the accident investigation, but to determine if a pilot could recover the aircraft after a hardover failure while flying IFR, i.e., no outside horizon reference. A flight test of simular magnitude would be dubious, because of the hazard of flying a test under IFR conditions, demonstrating the value of a simulator.

Hardover test were performed with reduced SCAS authority by varying the constants of the SCAS actuators in their respective subroutine package of the computer program. The baseline test were performed with approximately $\pm$ 12% pitch authority and was subsequently reduced to $\pm$ 10% and eventually to $\pm$ 8%.

The test concluded with the incorporation of a SCAS monitor system in the SCAS logic subroutine of the computer program. This was done to demonstrate the benefits of a SCAS monitor into the 214 helicopters. The principle of operation of the monitor is that it has a duplicate SCAS comparator network which in the event of a hardover will only disengage the faulty channel. Thus the hardover is nulified before it can change the helicopter attitude. Thus the remaining SCAS channels are left operating and the helicopter can safely be flown with only one SCAS channel off line.

SIMULATION RESULTS

The results of the baseline hardover simulator test depicting main rotor flapping versus pilot delay time and the respective pilot technique are shown in figures 7 through 10. The data shown in the aforementioned figures are derived from simulator flights that parallels the accident flight conditions. It should be noted that pilot technique number 5, is only applicable to lateral hardovers, i.e., flyout technique is designed for helicopters at high roll attitudes. Pilot technique number 4, was not flown for right roll hardovers because they did not prove effective in controlling previous hardovers.

PILOT DELAY TIME SEC	MAIN ROTOR FLAPPING - PERCENT				
	39 AND BELOW	40 TO 59	60 TO 79	80 TO 89	90 TO 100
1.0		3, 4, 5	1		2
1.5		3			
2.0		3	1, 4, 5		2
2.5			4	2	

FIGURE 7. PITCH UP HARDOVER RESULTS

PILOT DELAY TIME SEC	MAIN ROTOR FLAPPING - PERCENT				
	39 AND BELOW	40 TO 59	60 TO 79	80 TO 89	90 TO 100
.5		1	3		
1.0		4	1	2, 3	
1.5					1, 2

FIGURE 8. PITCH DOWN HARDOVER RESULTS

Figure 7 presents the results of a longitudinal hardover, pitch down, at the basic configuration. It should be noted that the maximum delay time was 1.5 seconds with pilot techniques number 1 and 2 resulting in the largest main rotor flapping. Technique number 2 was specially formulated as the technique which most probably resulted in the accident. Pitch down hardovers resulted in large pitch rates coupled with large roll rates to the right. The roll rate if allowed, built up to such on extent that it could not be controlled by the lateral cyclic stick/lateral rotor control power remaining. An adverse effect of pitch down hardovers is that it results in the aircraft entering a nose low attitude which results in an increase in airspeed and a decrease in altitude. In the event the 214 is engaged in high speed flight at a low altitude and a pitch down hardover is encountered the helicopter may exceed the Vne speed (Velocity Never to Be Exceeded). In addition, a significant loss in altitude may result, thus minimizing the altitude required for a safe recovery of the helicopter.

The data from pitch up hardovers are displayed in figure 8. Pitch up hardovers would result in a nose up altitude of the helicopter, and due to the visual system restrictions the pilot would loses his horizontal reference and would appear to be flying in a thick cloud cover. This made it necessary for the pilots to recover the aircraft by reference to their flight instruments. It is obvious that pilot technique number two resulted in the largest flapping and the minimum pilot delay time, clearly the worse condition.

PILOT DELAY TIME SEC	MAIN ROTOR FLAPPING - PERCENT				
	39 AND BELOW	40 TO 59	60 TO 79	80 TO 89	90 TO 100
1.5	4, 5	1, 2			
2.0		1, 3	2		
2.5		4, 5			
3.0		3	5	2	
3.5			1		

FIGURE 9. LEFT ROLL HARDOVER RESULTS

PILOT DELAY TIME SEC	MAIN ROTOR FLAPPING - PERCENT				
	39 AND BELOW	40 TO 59	60 TO 79	80 TO 89	90 TO 100
1.0	1				
1.5	3, 5	2			
2.0	1, 3	5	2		
2.5		3	1, 5		2

FIGURE 10. RIGHT ROLL HARDOVER RESULTS

Roll hardovers were generally less severe than the pitch or longitudinal hardovers. Figures 9 and 10 clearly depicts that pilot technique number two resulted in an increase in main rotor flapping

over all other techniques tested. The maximum delay time was approximately 3 seconds for a right roll hardover and 3.5 seconds for a hardover resulting in a left roll. Flight test have shown that yaw hardovers are very mild and due not result in large yaw rates or altitude excursions, and consequently, did not warrant investigation.

The data from the other testing is not shown but will be discussed in general terms. The effects of changing the center of gravity from the baseline aft c.g. to a forward c.g. of 134.6 inches was minimal on the main rotor flapping and delay time. Airspeeds of 100, 120, 140, and 160 knots were tested for all single axis hardover combinations. The results showed that there was no noticeable increase in flapping due to the increased airspeed, except for the case of 160 knots pitch down hardovers which showed a slight increase.

The hardovers that were induced under simulated IFR resulted in no difference in response. This is to say that flapping and maximum time delays are comparable to those of the baseline testing shown in figure 7 through 10.

As previously mentioned, the flight control systems of the 214 helicopter ws mathematically modeled in such a way that the SCAS control gains could be adjusted to investigate their effects on the helicopter handling and stability characteristics during normal flight and hardover failures. During normal trimmed flight, the pilots noted no qualitative difference in helicopter handling qualities with the reduced SCAS pitch authority. The maximum delay times were not effected by varying the SCAS authority, however, the pitch down flapping response was reduced approximately 20 percent when the SCAS authority was reduced to 8 percent.

The results of the SCAS monitor testing showed that the monitor would disengage the failed SCAS channel and center the series actuator. The monitor reduced the main rotor flapping appreciably and made the pilot's job of correcting the hardover minimal.

ACCIDENT DETERMINATION RESULTS

Throughout the testing, it was impossible to achieve a pitch down and left roll altitude from either single or dual hardovers. This is the attitude observed during the accident. The observer was confidient about the pitch down attitude while being hesitant about the left roll attitude. Pitch down hardovers with delays of 1.5 seconds resulted in a nose low attitude and when pilot techniques 1 or 2 were used for recovery excessively high main rotor flapping resulted. However, pitch down hardovers in the simulator would

result in a large roll attitude to the right which was contrary to the observed attitude. This can be explained by the fact that the observer was hesitant about the roll attitude and that the pitch down right rod coupling of the simulator was exaggerated over that of the real helicopter. Predominantly, large flapping would result if the hard-over was removed inadvertently at the start of the pilot recovery maneuver, i.e., pilot technique number 2.

Although it was impossible to ascertain conclusively the exact cause of the accident, it is most probable that the accident was caused by a pitch down hardover during flight testing, and its inadvertent release prior to recovery.

CONCLUSIONS

From this program, one can conclude that future Army real-time pilot-in-the loop simulations have considerable merits as a research and development tool. A simulation of the magnitude as outlined in this paper would be extremely costly and time consuming if it had to be performed by flight testing. Also there is considerable pilot risk in flight testing the failure modes of the helicopter control system. Thus simulators are ideal for test that involve a degree of risk during flight testing. A simulator program is cost effective because they require less people to be involved, require less time, and are not affected by weather (can fly 24 hours a day if need be and do not have to wait for good flying weather).

The ability to modify the math equations to represent different test conditions, e.g., gross weight, center gravity, airspeed, etc., and control system configurations of the basic helicopter, as demonstrated by this will result in the simplification of the airworthiness/qualification process of future Army helicopters. This will allow Army assessment of a new/development helicopter's handling qualities and performance, while the helicopter is still in the development phase. The pilot can perform maneuvers in the simulator that are physically prohibited during normal flight testing of development helicopters.

RECOMMENDATIONS

As future follow on, consideration should be given to the following recommendations for piloted simulation as a development tool:

1. Modification of math model to incorporate gust and turbulence inputs into the simulation, thus allowing the pilot to assess the effects of a turbulent environment.

2. Piloted simulation can be used to effectively evaluate simulated night flight and the related effects on pilot workload.

3. Through modification to the basic math model, different configurations of the basic aircraft can be represented and tested. An example would be the effects of a stretched airframe, improved engines, and different control surfaces. Not only can they be evaluated but their design can be optimized without having to build actual hardware.

4. The simulator could be used to effectively evaluate handling quality requirements and military specifications more effectively than previously done in flight test programs. (10 & 11).

5. Piloted simulation could be accomplished with actual flight control hardware interconnected with the simulator to verify and qualify equipment operation (12 & 13).

6. The simulator is an ideal tool to perform helicopter failure analysis, e.g., control system failure, engine failure, loss and center of gravity changes (store jettison), assymetric c.g. effects, etc. Simulator would be used to evaluate the effects of the above failures and to optimize a helicopter design that would minimize the failure.

7. Human factor evaluation of new equipment and its installation location can idealy be accomplished on a simulator with its ease of equipment change.

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